Title: MMS Mission Study		**Date of RFO:**	January 27, 2000
Type: Mission Specific		**Date Responses are Due:**	February 11, 2000 3PM, Eastern
Ordering Clause: I.A.7		**Required Completion Date:**	100 days

Summary: The Magnetospheric Multiscale (MMS) Mission Study is an accommodation assessment to determine the range of suitable concepts. Of primary importance are fitting 5 spacecraft in a single launch vehicle fairing, methods of achieving mission objectives and cost. Cost is a significant driver for this mission. Multiple study awards are anticipated. The studies will focus on meeting performance requirements, heritage, and price justification. The government anticipates awarding up to six delivery orders for this study. If not selected for the study delivery order, RSA contractors may elect to perform the study on their own since any future competition for a MMS mission will not be limited to those contractors receiving a study.

List of Attachments:

A MMS Mission Study Statement of Work
B MMS Mission Description
C MMS Mission Spacecraft Requirements
D MMS Mission Instrument Suite Specification
E Deliverable and Price

Offer Format and Instructions:

a) The offeror shall submit a narrative that outlines their concept for implementing the MMS Mission. The submission must address the proposed approach to fulfilling the requirements of the Statement of Work and performance specification. Options or deviations from the core bus must show a credible path for on-ramp by 2003.

b) Offers are to contain no more than two proposed payment milestones and associated completion criteria associated with the study effort.

c) Page limited to no more than 10 pages in not less than 12 point font. The font in tables and figures are to be no less than 8 point. Fonts may be either Times New Roman or Ariel, embedded graphics are to be JPEG or GIF format only and not exceed 100Kb each.

d) Any offer containing a virus will not be opened or evaluated.

e) Offers must be submitted by the above date and time at the RSDO Web Site (http://rsdo.gsfc.nasa.gov)

Evaluation Factors and Relative Importance:
The evaluation team will use the following factors in selection and award.
The Cost/Price Factor is significantly less important than the combined importance of the Mission Suitability Factor and Relevant Experience and Past Performance Factor. As individual Factors, the Cost/Price Factor is less important than the Mission Suitability Factor but more important than the Relevant Experience and Past Performance Factor.

a) Suitability for the MMS Mission
NASA will evaluate the offeror's proposed approach and concept for accomplishing the activities reflected in the SOW and meeting the performance specifications and mission requirements. Any new heritage or availability, beyond that specified in the IDIQ contract, of spaceflight components, subsystems, and systems will also be evaluated.

b) Price
The reasonableness of the proposed price shall be evaluated including consideration of the funding limitation of $120K per delivery order.

c) Relevant Experience/Past Performance
Any new relevant experience or past performance beyond that submitted in the RSA proposals will be examined.

Appendix A

MMS Mission Study

Statement of Work

Magnetospheric Multiscale
Mission Study
Statement of Work

1.0 Introduction

This Statement of Work specifies the work to be performed by the contractor in support of the Magnetospheric Multiscale (MMS) Mission Study. This activity is intended to begin the process of getting industry input to mission planning and architecture development leading up to a competitive selection process for implementing the MMS mission. Following this short study activity the Project plans, budget permitting, to embark on more detailed study activities (a separate procurement action), which would include interaction with instrument team representatives to further detail the instrument, interfaces.

2.0 Work to be Performed

The contractor shall demonstrate the feasibility of accommodating five identical sets of the MMS science instruments, identified in Appendix C of this document, on five identical spacecraft within the Delta 7925 3 meter fairing.

2.1 Purpose

The purpose of this study is to demonstrate feasibility of accomplishing the proposed mission. This shall be accomplished by showing a mission concept that has mass, propellant and power margins commensurate with this early phase of development and by showing a fabrication, assembly & test concept that enables cost effective multiple spacecraft builds and instrument integration & testing. The spacecraft subsystems shall be detailed and any modifications to standard or 'catalog' designs should be highlighted. In addition to mid-term and final presentation meetings, the contractor shall deliver two products: 1) A 3-D CAD model in an industry standard format of sufficient detail to allow the NASA teams to do coarse fit and rough FOV studies. 2) A final report sufficient to assist in the planning, formulating and advocating the MMS mission.

2.2 Assumptions for This Study

During the course of the study the contractor shall assume:

a. Launch date for the mission is 12/05.
b. Implementation phase (phase C/D) begins 36 months prior to launch.
c. All satellites shall be launched by a single launch vehicle (7925 class).
d. The launch vehicle will deposit the 5 spacecraft payload, maximum mass of 1540 kgs, into a 1.2 x 12 RE orbit at 28° inclination. The spacecraft shall provide the means to provide the ΔV of 326 m/s needed to perform the necessary plane change to arrive at the Phase I orbit.
e. The mission description is as documented in Appendix A.
f. The preliminary spacecraft specification is provided in Appendix B
g. The instruments for the mission are those described in Appendix C.

2.3 Accommodation Studies

The contractor shall develop a mission architecture that supports the mission as described in Appendix A. Of particular interest, the contractor shall address the following:

a. Fitting 5 spacecraft in a single launch vehicle fairing with adequate access space, clearances & margins.
b. Given your approach to item "a" above, describe your separation strategy for the initial separation from the launch vehicle of the 5 spacecraft and/or subsequent individual spacecraft separations.
c. Adequate instrument accommodations with clear FOVs during science operations
d. The baseline mission requires the satellite to spin at 20 rpms and still maintain the spin axis knowledge $\leq 0.1°$. Provide details of how that requirement is met - identify the hardware necessary and reference past hardware performance if applicable. If the

baseline was to change to a 6 rpm spin rate would the hardware chosen to meet the <0.1° knowledge requirement change? Would the different field of view for the hot plasma instrument operating on a 6 rpm spinner cause accommodation problems (see appendix C for details)?

e. The current baseline is to have the Inter-spacecraft Ranging and Alarm System, IRAS (see appendix C for details), provided GFE to the spacecraft vendor during implementation, much like the instruments are provided. As part of this study provide your alternative concept, if any, for accommodating the IRAS requirements as part of the vendor spacecraft architecture, essentially making it part of your existing spacecraft subsystem, like the communications subsystem, or a stand alone subsystem.

f. Provide a detailed description of the fabrication, assembly, integration and test process for the five spacecraft constellation.

g. If new technology is deemed necessary to meet this mission's requirements, provide a plan on how the contractor would introduce the new technology; including schedule, milestones, etc.. For this study the spacecraft bus to be baselined does not have to be in the catalog; but the mission will evaluate the reasonableness that the bus will exist in the catalog at the time of spacecraft buy.

2.4 CAD Models

2.4.1 Provided: GSFC shall provide Models (DXF, STEP or IDEAS) of the launch vehicle fairing and Hot Plasma and Energetic Particle instruments, including their field of view, to the selected vender(s) to assist in the accommodation activities. Models for the magnetometer and electric field instrument are not included since the configuration is highly dependent on your accommodation scheme. Descriptions of those instruments are included in Appendix C.

2.4.2 Delivered: The contractor shall provide a CAD model (30-40 meg max in a mutually agreed format) of their proposed spacecraft concept at the conclusion of the study activity. The model shall reflect your concept of the five fully integrated spacecraft successfully encapsulated into the launch vehicle fairing.

2.4.3 Transfer of models will be via a GSFC established FTP site.

2.5 Final Report

The final report shall include descriptive text to assist in conveying the technical approach to the mission. The final version of the final presentation in viewgraph form is sufficient providing it includes facing page text, which will provide that descriptive information. Also, included in the final report shall be a rough order of magnitude of mission costs (to a level 3 WBS) through 30 day on-orbit checkout and delivery and a proposed schedule. This information shall be used only for planning and formulation purposes.

Appendix B

MMS Mission Description

MMS Science And Mission Architecture Assumptions

Purpose:
This document is an attempt to capture the science goals of the MMS mission and their impact on the mission architecture. This is not an attempt to provide a detailed description of the science goals but rather identify the general regions of interest in the magnetosphere and the ramifications on the mission architecture of investigating those regions.

General:
The MMS mission is designed to investigate the Earth's magnetosphere with special attention paid to the neighborhood of the magnetopause and plasma sheet. The mission has been broken up into four phases. Phase 1 focuses on the predicted subsolar magnetopause reconnection region and the substorm current disruption region of the near magnetotail. Phase 2 focuses on investigating the magnetotail at distances up to 30 Re with special interest in the substorm reconnection region. Phase 3 focuses on investigating the distant magnetotail to approximately 120 Re, while rotating the orbit plane to become perpendicular to the ecliptic. Phase 4 is an investigation of the magnetopause from high to low latitudes from a 90° inclined orbit that skims along the boundary of the magnetosphere.

It is assumed that the spacecraft 'formation flying' shall be focused such that the tetrahedron is formed in the apoapsis region of the orbit (formation flying is used loosely here since no action is taken by the spacecraft to maintain the formation).

Mission Architecture:
<u>Phase 1:</u>
The nominal subsolar distances to the bow shock and magnetopause are 15 Re and 10 Re, respectively. With an apogee of 12 Re during Phase 1, the MMS spacecraft will sample the magnetosheath region between these two boundaries, and as upstream conditions vary multiple boundaries crossings will occur. An additional area of scientific interest during this phase is the study of magnetic substorms on the night side of the magnetosphere. These two scientific objectives will dictate the minimum duration of this phase. Assuming that in order to study substorms the mission orbit apogee needs to pass through the tail, it will take 9 months for the mission orbit to ‘traverse’ from launch near 0300 local time, through the tail and then past the subsolar magnetosphere. See the clock diagram in figure 1.

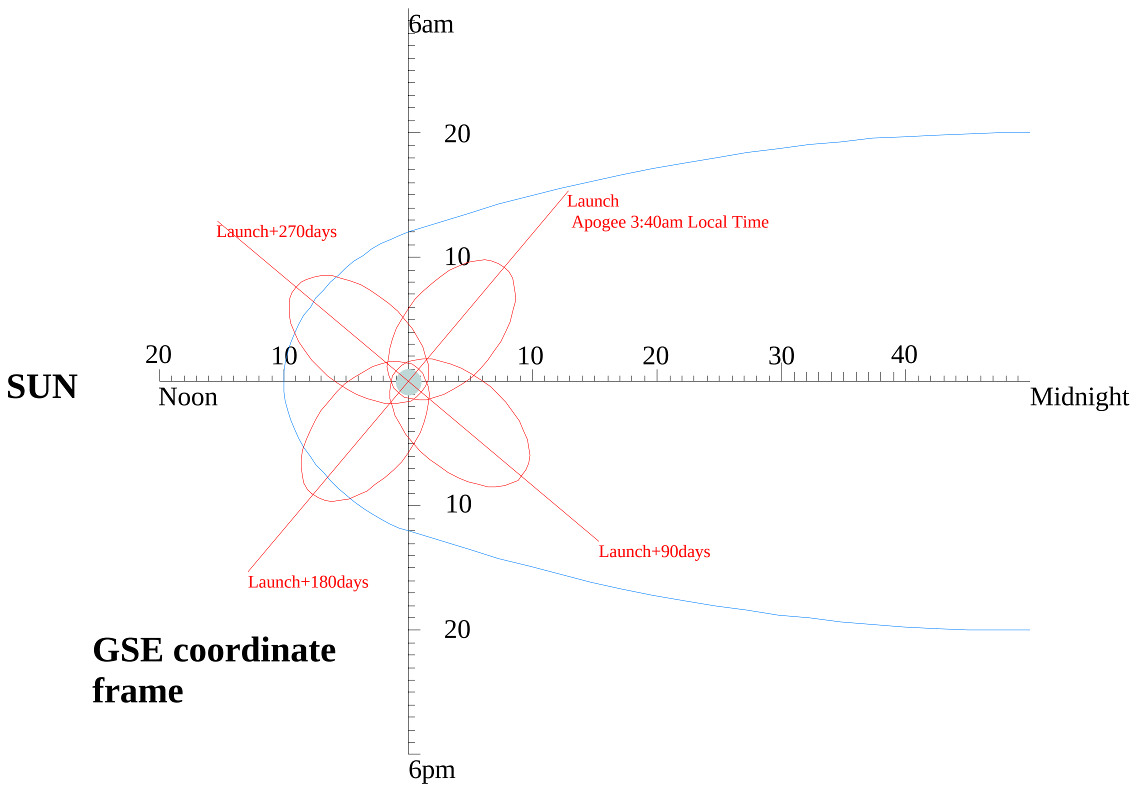


Figure 1; Phase 1 Orbit History Diagram

Since the earth's magnetic field dominates at close distances the phase 1 orbit has been set up so that its line of apsides is in the magnetic equator. The orbit period will be adjusted to be exactly synchronous. The argument of perigee for phase 1 is 0. This results in apogee & perigee being placed over the intersection of the magnetic & earth's equators i.e. at ~ 15E or 156W.

Phase 2:

In order to explore the substorm formation region in the magnetotail, the mission orbit apogee will need to rise. Figure 2 shows a clock diagram illustrating this transition.

As the apogee is raised, MMS will explore the dawn flank of the magnetosphere, with excursions through the low latitude boundary layer (LLBL) back and forth across the magnetopause. As the orbit traverses the magnetotail, the scientific requirement is to remain close to the plasma sheet for as long as possible. Balancing the requirement to be close to the plasma sheet with engineering constraints to avoid long eclipses dictates a seasonal time constraint on the optimal orbit. To avoid long shadows, and maximize plasma sheet occupancy, it is best to cross the midnight meridian at a time when the plasma sheet is displaced from the GSM equatorial plane, i.e. when the tilt angle is largest in magnitude around the months of June or December, see Figure 3. If we set up the mission orbits such that either June or December corresponds to the midnight position in figure 2 we can then step backwards in time to determine our launch windows.

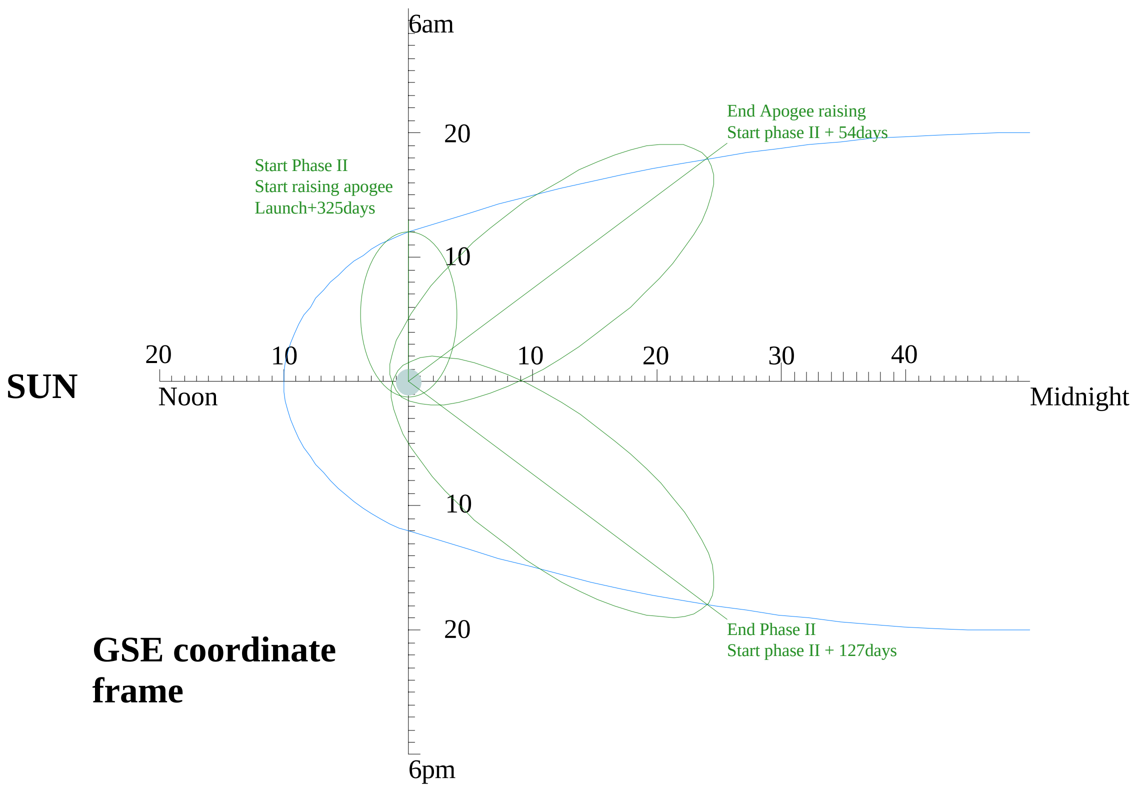


Figure 2; Phase 2 Orbit History Diagram

Following this assumption, it is assumed that the orbit plane, during this latter portion of phase does not need to be coplanar with the magnetic equator only that the orbit apogee passes through the central plasma sheet when apogee is at midnight. See figure 4 for details.

Phase 3:
This phase is primarily an orbit transition phase utilizing the gravitational pull of the moon to increase the perigee and apogee distances and induce a ~90° orbit plane change. This is accomplished with 2 sets of double-lunar-swingby (DLS) maneuvers. In order to complete all phases within the planned mission lifetime, the duration of the DLS will be no longer than one month. It is assumed that these maneuvers will be accomplished on the night side of the orbit so that during the maneuver the spacecraft will make deep passes (~100-120 R_E) into the tail region to conduct science investigations of that region.

Phase 4:
This final mission phase involves taking measurements of the high latitude magnetosphere in an orbit that, at the beginning of the phase, is inclined 90° with respect to the ecliptic. On the dayside, MMS will sample the cusp/cleft regions as well as subsolar and high-latitude reconnection as the Interplanetary Mangetic Field (IMF) orientation changes. On the nightside, north-south cuts of the plasma sheet at different distances will provide information on the dynamics of the plasma sheet. Finally, traversals of the high latitude boundary layer will investigate transport of solar wind plasma across the magnetopause.

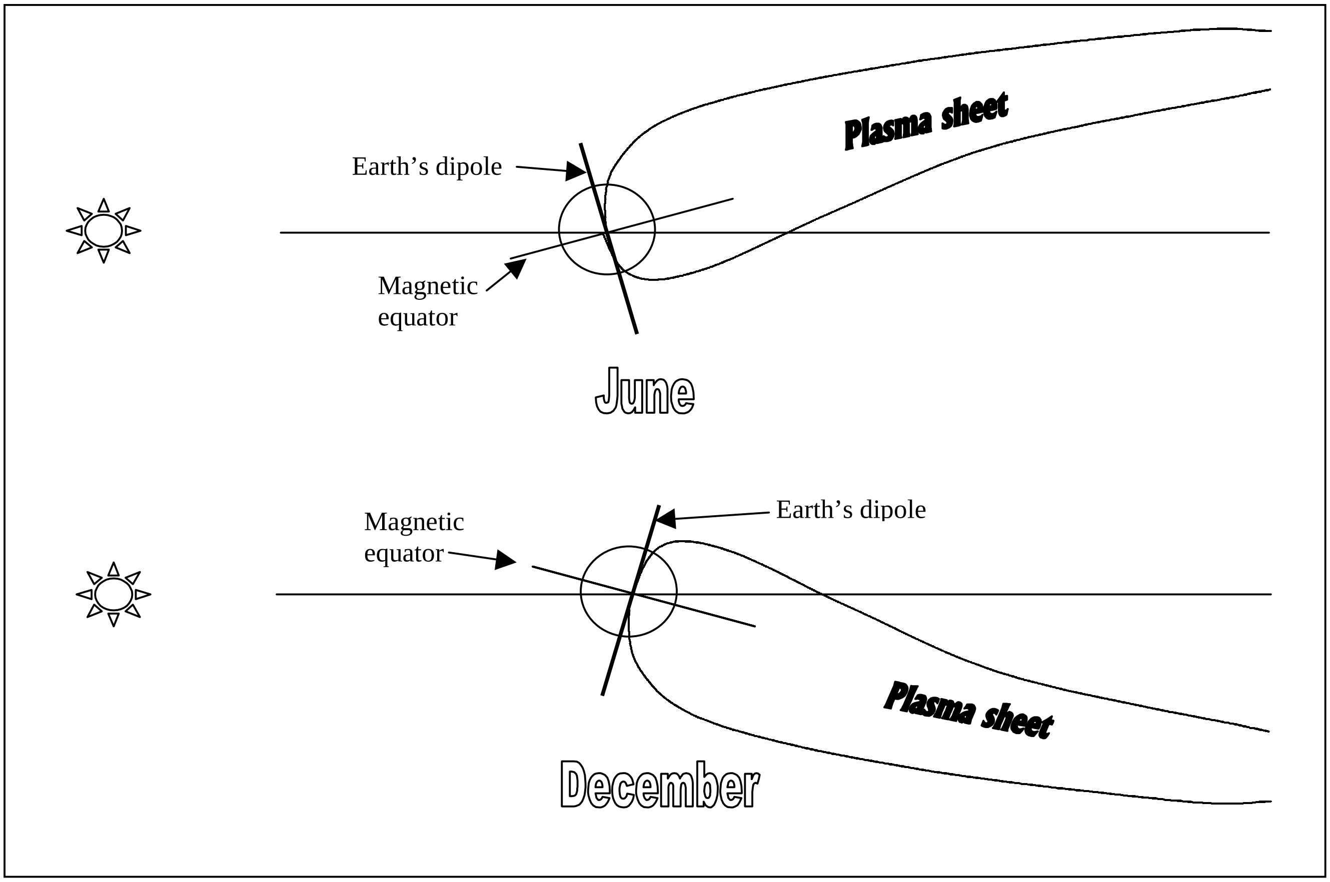


Figure 3; Plasma Sheet Orientation as a Function of time of Year

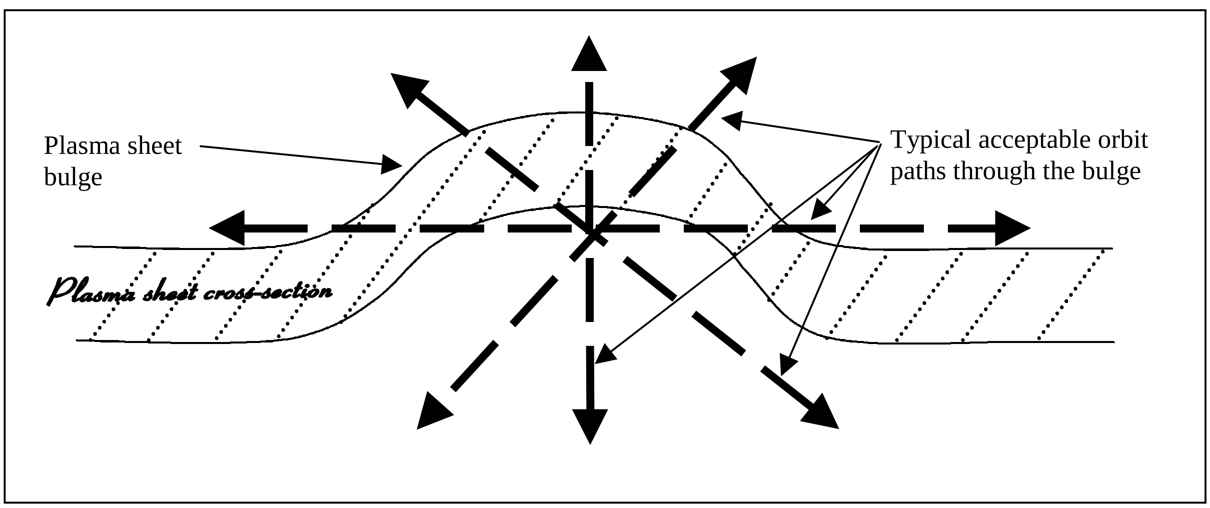


Figure 4; Cross-sectional View of Plasma Sheet from Earth looking Anti-Sun

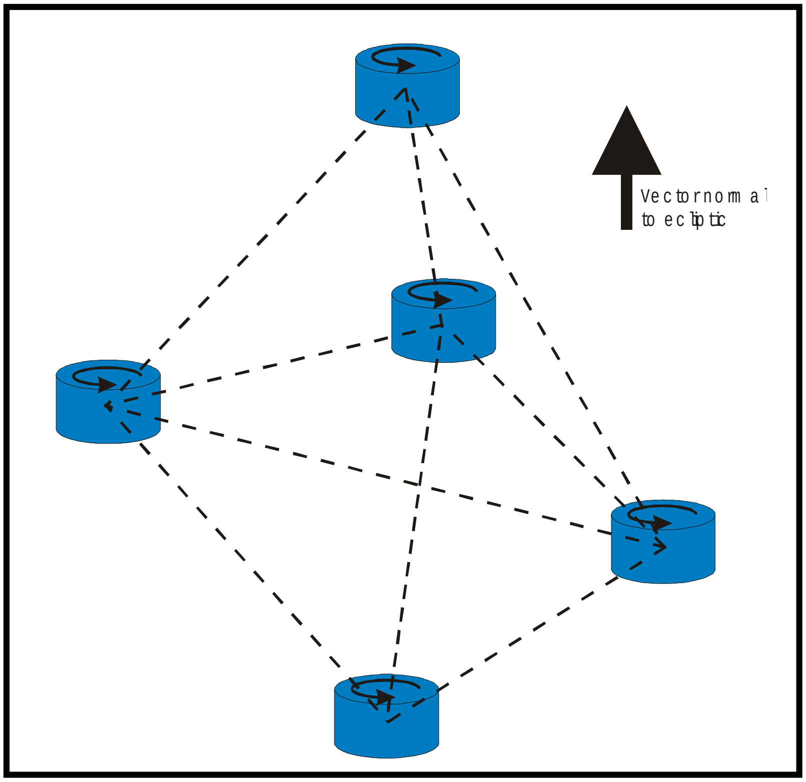
Appendix C

Magnetospheric Multi-Scale Mission

Spacecraft Requirements

1 Introduction

The MMS mission consists of five identically spinning spacecraft acting as a single probe, flying such that a hexahedral configuration like that shown in figure 1 will form at apogee with inter-spacecraft spacing of that configuration varying from less than 10km to several Re throughout the mission life. Each spacecraft will contain an identical set of instruments.



<u>Figure 1 – Hexahedral Configuration</u>

Onboard propulsion will allow the orbits of the MMS "probe" to have four separate mission phases, covering almost the entire magnetosphere, from near-Earth equator to the magnetotail. In each phase the 5-s/c "probe" will dwell at apogee in key boundary regions of magnetic reconnection and energy conversion.

2 Multi-probe Spacecraft

Five identically instrumented spacecraft will comprise the MMS mission.

2.1 Launch vehicle Compatibility

2.1.1 Vehicle

The mission design assumes that all 5 spacecraft will be launched from a single Delta 7925 with a 10' fairing.

2.1.2 Mass to orbit

The Delta 7925-10 can launch 1540kg to the injection orbit. The PAF mass is not to be included in the 1540kg but what ever structure is needed to stack and individually release the five spacecraft shall be.

2.2 Definition of the orbit

Inclinations are relative to the earth's equator. The orbits and the apogee & perigee distance from the center of the earth are given in units of earth radii.

2.2.1 Phase 0 (launch)

The launch vehicle inserts all 5 spacecraft into a 12Re x 1.2 Re 28.5° orbit. The spacecraft are ejected spinning from the Delta third stage. The time of launch is set so that apogee is approximately at local midnight.

2.2.2 Phase I

The probes change orbit plane to 12Re x 1.2 Re 10° orbit. They stay in this orbit for ~10 months. The apogee yields long dwell times in the near tail region. The probes initially form a double tetrahedron (hexahedron) with side 10km at apogee. During phase 1 the tetrahedron spacing is increased to 0.1Re.

2.2.3 Phase II

The probes change to 30Re x 1.2Re orbit in stages. The goal is to keep apogee on the dawnside of the magnetopause while raising the apogee from 12 to 30 Re as the magnetic local time changes. The probes form a hexahedron with the sides varing between 0.01 Re to 1 Re. Apogee remains in the midnight side of magnetosphere as the magnetic local time of the apogee changes from ~1000 to ~0400 hours. They stay in this orbit for ~4 months.

2.2.4 Phase III

Phase III Double lunar flyby changes the orbit inclination to ~90°. This phase lasts approximately 3 months.

2.2.5 Phase IV

Phase IV changes to 50Re x 10Re polar orbit with apogee in the equatorial plane. In this orbit perigee skims the magnetopause from cusp to cusp. They stay in the phase until the consumables are exhausted (expectation is approximately 7 months.)

2.2.6 Summary of Mission Phases

Parameter	Injection/ Separation Phase 0	Phase I	Phase II Apogee follows Magntotail	Phase III Double Lunar Swingby	Phase IV
Orbit (Re)	1.2 x 12	1.2 x 12	1.2 x 30	8 x ~120	10 x 50
Orbit Period	24hr	~24hr	3.6day	~30day	9.6day
Inclination	28°	10°	10°	N/A	90°
Arg of Perigee	0° (daylight)	0°	Apogee 1000 to 0400 hr local time	TBD	0°
Max Ecipse	1hr	1hr	*	*	1.2hr
Eclipses per year	N/A	255	TBD	TBD	4-6
Satellite separation	<10km	10km to 0.1Re	0.01Re to 1Re	1Re to 10Re	10km to 1Re
Phase duration	hours	~0.8 yr	~0.3 yr	~0.25 yr	~0.6 yr

* The mission design and launch time constraints will limit the worst case eclipse duration to < 2hr.

2.2.7 Delta V Budget estimates

Maneuver	**Phase**	**Nominal dV (m/s)**
Inclination change	Separation/Phase 0	326
Apogee Adjust	I	277
Apogee Adjust	II	110
Double Lunar Swingby/Phase IV initialization	III	90
Perigee maintenance	I, II, III, IV	48
Launch window/Arg of Perigee adjust	Launch/0	35
Launch dispersion	Launch/0	12
Tetrahedron development	I, II, IV	120
Total		**1018**
Capability		1100
Margin		82

2.3 Design Lifetime

The mission science requires the virtual 'sensor' consist of at least 4 spacecraft. The baseline mission architecture consists of 5 spacecraft. This architecture allows for implementation of an aggressive spacecraft design philosophy to maximize design simplicity and minimize redundancy and still ensure meeting science goals. The overarching reliability goal is to have at least 4 spacecraft still operating after the two year mission life time.

The expendables for this mission are to be sized for a dV of at least 1100m/s.

2.4 Spacecraft Attitude Control.

A spin-stabilized spacecraft is assumed with the following parameters:

2.4.1 Spin axis

Spin axis offset from normal to ecliptic plane by 5°. Offsetting the spin axis by 5° prevents the body of the spacecraft from shadowing the electric field sensors.

2.4.2 Spin rate control

Spin rate controlled to 20rpm (±0.2rpm)

2.4.3 Spin axis knowledge

Spin axis knowledge (post-processed) with respect to inertial space < 0.1°, spin phase knowledge (post-processed) < 0.1°. See also discussion of magnetic field instrument.

2.4.4 Orbit determination:

Knowledge of individual spacecraft position is 100km.

2.4.5 Spacecraft stability

The spacecraft shall be stable in all mission phases. Prior to the deployment of the electric field booms the spin-to-tumble inertia ratio shall be > 1.04. The axial and spin plane booms will dominate the dynamics of the spacecraft after deployment, for analysis purposes the axial booms may be assumed to have the properties as described in paragraph 1.4 of the Instrument Specification.

2.5 Electrical Power

Instrument orbit average power is given in the instrument summary section of the MMS Instrument Suite Specification table 1. It is anticipated that extended eclipses will be encountered during phased I, II, III & IV for the MMS spacecraft. A detailed analysis has been performed of eclipse duration throughout these phases. Eclipse duration will be limited to a maximum of 120 minutes and instruments will be assumed to remain powered during eclipses.

It may be assumed that the instruments will work with bus voltages within the range of 22V-35V.

2.6 Communications

2.6.1 Uplink/Downlink Frequency

The spacecraft shall use an X-band frequency for uplink and downlink.
Command link shall be maintained at maximum apogee.

2.6.2 Groundstation stratagy

Commercial 11m groundstations will be used for phases I & II. DSN 34m High Efficiency (HEF) groundstations will be used for phases III & IV. In order to minimize groundstation cost, the spacecraft shall have the capability to store up to 14days of science data without loss and downlink the data at a rate of at least 1 Mbps but less than 2.2Mbps. The intent is to store data until the range to the groundstation is short enough to allow the data to be transmitted at a high rate.

2.6.3 Spacecraft EIRP

The spacecraft shall be capable of transmitting at least 20dBW in the direction of the earth. The transmitter shall be capable of transmitting continuously for at least 4hours.

2.6.4 Science data/Commanding Volume

The instrument complement will generate ~2 Gbits per day per spacecraft (see table 1 of the instrument Suite Specification.

Commanding for the instruments shall be 100 bytes per instrument per day for each spacecraft.

2.6.5 Tracking

The strategy for tracking the spacecraft and determining their orbit is currently under study. The options being considered are to use two-way Doppler or to require the spacecraft to fly high stability oscillators and use one-way Doppler. Two-way Doppler is the current baseline with a requirement of 100km for normal operations.

2.6.6 Command and Telemetry Format

The uplink and downlink format shall be Consultative Committee for Space Data Systems (CCSDS) Advanced Orbiting Systems (AOS) format.

2.7 Environments

2.7.1 Radiation Environment

The anticipated total radiation dose, without margin, for the MMS mission is shown in Figure 2.

2.7.2 EMC

Since the instruments measure very low levels of plasma energy the spacecraft must not disturb the surrounding plasma. The spacecraft exterior surface shall be an iso-potential surface with no

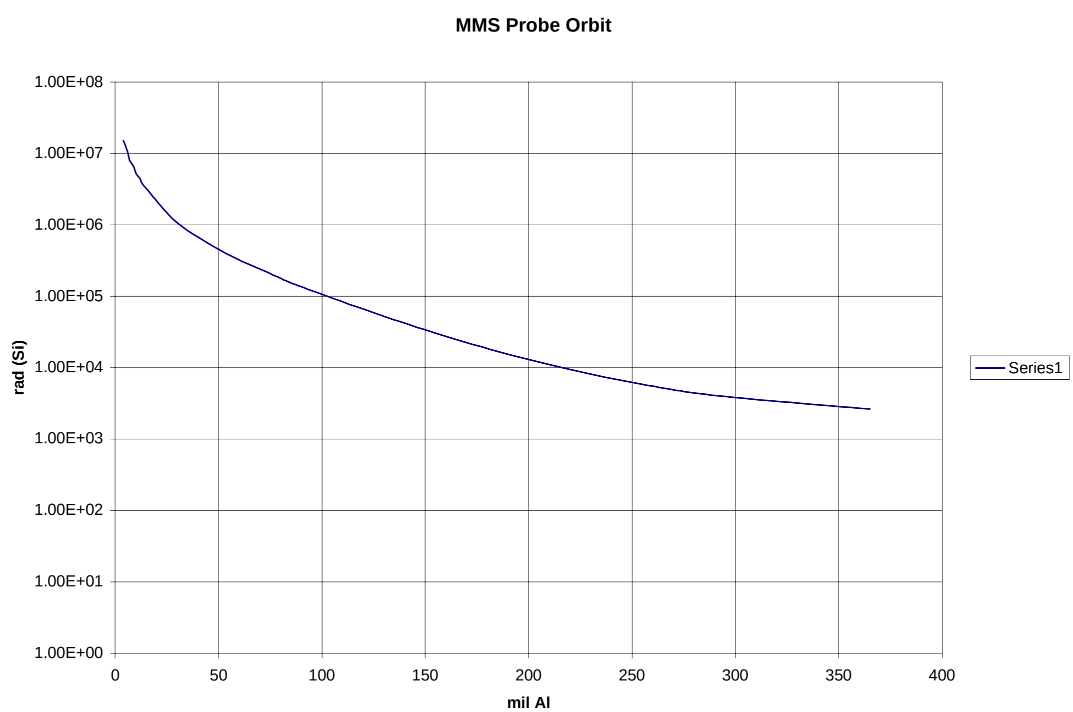


Figure 2 MMS Total Dose Curve

point of the exterior surface more than 1Volt different than any other point on the spacecraft. This requirement applies to all surfaces including blankets and solar arrays.
The magnetic compatibility requirements are described in Magnetometer instrument section of the MMS Instrument Suite Specification (section 1.5.1)

2.7.3 Contamination

The Hot Plasma instrument and the Energetic Particle instrument are susceptible to particulate contamination and must be under high purity nitrogen purge until launch.

2.8 Mechanical

The spacecraft design shall be compatible with the launch environment for a Delta II 7925-10 and support the instrument allocations identified in the MMS Instrument Suite Specification. The mechanical design shall accommodate the fairing volume, center of gravity location, and stiffness requirements for the Delta II. The mechanical subsystem shall have provision to provide dedicated high purity nitrogen purge to the Hot Plasma and Energetic Particle instruments of all spacecraft through launch.

The mechanical layout of the instruments and spacecraft shall provide unobstructed fields of view for the Hot Plasma and Energetic Particle instruments.

Appendix D

Magnetospheric Multi-Scale Mission

Instrument Specification

1 Instruments

1.1 Instrument Summary

While the instruments have not been selected for the MMS mission, the science definition team has suggested a candidate list of instruments. For the purposes of this study the list of instruments will be as shown in Tables 1 & 2.

Table 1

Parameter	Magnetometer	Hot plasma detector	Energetic Particles	Electric Field	Totals
Mass (kg)	1.5	8.0	2.5	17.0	37 kg
Quantity	1	2	1	1	
Operation Power (W)	1.2	7.0	2.0	15.0	32.2 watts
Data Rate (Normal)	5 kbps	6 kbps	2 kbps	5 kbps	24 kbps
Data Rate (Burst)	5 kbps	32 kbps	2 kbps	65 kbps	104 kbps
FOV (deg)		10° x 360° (1)	10°x160°		
Size (cm)	18x10x8 (Elec. box)	20.3x20.3x25 (instr & elec. Box)	11x11x10 (instr & elec. Box)	See section 1.4 & see figure 4	

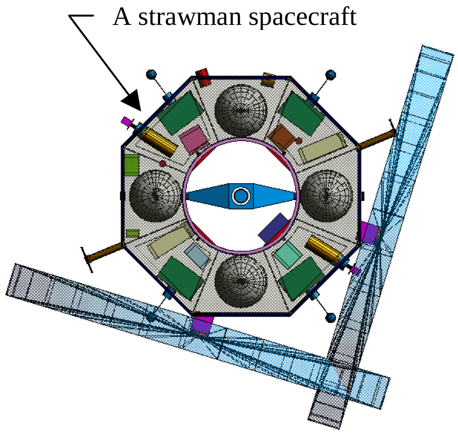
(1) For non-scanning version of the instrument. See description of instrument.

Table 2

Parameter	IRAS
Mass (kg)	1.2kg electronics + 0.8kg antennas
Operation Power (W)	5 (avg), 15 (peak)
C&DH storage Rate	100bps
C&DH transfer rate	500bps
Size cm (LxWxH)	12 x 12 x 4

The Interspacecraft Ranging and Alarm System (IRAS) shall be considered a GFE'd instrument for the purposes of this document.

1.2 Hot Plasma Instrument



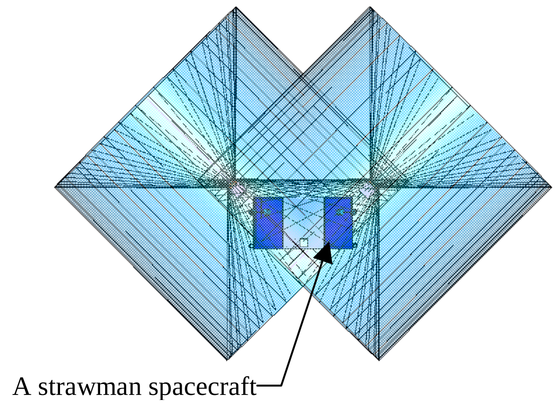


Figure 1, Non-scanning 20 rpm- Hot Plasma FOV

Figure 2, Scanning 6rpm – Hot Plasma FOV

The Hot Plasma instrument is used for measuring the number, density and energy for particles in the range of 1eV to 30eV. The instrument will have a geometry factor of 0.01 cm^2-sr-eV/eV for ions and 0.005 cm^2-sr-eV/eV for electrons, an energy resolution of 4 ($E/\Delta E$) and an absolute accuracy of 5-10% with a relative accuracy of 1% between spacecraft. The instrument can be built in one of two configurations: a non-scanning instrument with a field of view (FOV) of 10° x 360° (shown in figure 1) and a scanning version with an instantaneous field of view of 10° x 360° but the beam can scan ±45° (shown in figure 2). Note that the spacecraft shown is for illustrative purposes and is not meant to imply a preferred spacecraft configuration.

The two non-scanning instruments (per spacecraft) can provide 4π steradian coverage at 0.75seconds time resolution if they are mounted as shown in figure 1(~ 90° apart). The time resolution is inversely proportional to the spin rate. For slower spinning spacecraft, an electrostatically scanning instrument has been developed which can provide the time resolution independent of spin rate. To provide 4π steradian coverage, two of the scanning instruments are mounted at 180° to each other as shown in figure 2. In each case the instrument sensor body is approximately a cylinder 8inch diameter and 8inch long. Impingement into the Hot Plasma instrument FOVs by the wire booms of the electric field instrument is deemed acceptable.

The non-scanning instruments have been baselined for the MMS mission. In the event that the spacecraft is not capable of meeting the pointing requirements at 20rpm it is possible to use the non-scanning instrument at 6 rpm.

This instrument will require a constant high purity nitrogen purge from integration to launch. A purge port will be provided at the instrument interface.

It can be assumed that the instrument is capable of a Mil-Std-1553 or other high level interface.

1.3 Energetic Particle Instrument

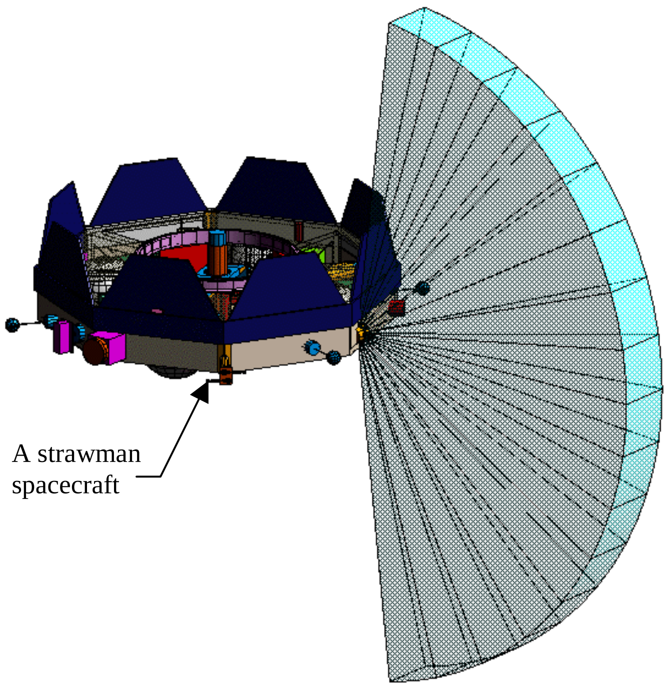


Figure 3, Energetic Particle Field of View

The energetic particle instrument measures number density, angle, energy and mass/charge for particles in the range 30keV-300keV. The instrument has a field of view of 10° x 160° as shown in figure 3 (the spacecraft shown is for illustrative purposes and is not meant to imply a preferred spacecraft configuration.) The instrument uses the spin of the spacecraft to give almost complete coverage. There is, however, a 20° hole in the coverage around the spin axis. This instrument will require a constant high purity nitrogen purge from integration to launch. A purge port will be provided at the instrument interface. It may be assumed that the Energetic Particle Instrument has a Mil-Std-1553 or other high level interface.

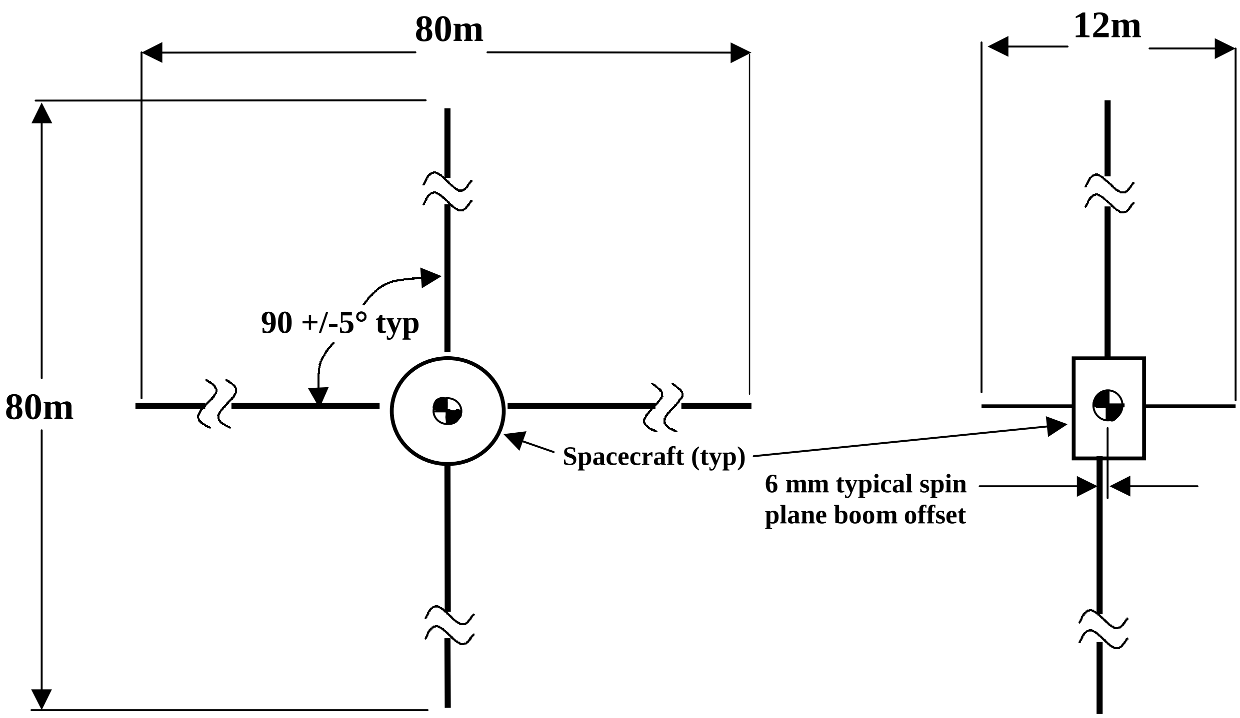


Figure 4; Electric Field Instrument

1.4 Electric Field Instrument

The electric field instrument consists of spin plane booms, axial booms and an associated electronics box.

1.4.1 The axial booms (right side of figure 4)

- Each axial boom has a 6m deployed length and less than .35 m stowed length along the deploying axis (not including the sensor sphere). Stowed volume less than 0.0075 m^3.
- Capable of supporting a single 0.15 Kg mass (10 cm diameter sphere) at the tip end and a cable bundle along the entire length. The minimum (fixed base) first mode bending and torsional frequency is 1 Hz.
- Deployed boom tip precisely positioned and stable within a full cone angle of 0.2° with respect to a known reference plane while under the influence of thermal effects consistent with a low earth orbit.
- Boom and deploying mechanism must be less than 2.5kg.
- Axial booms may be packaged end-to-end or side-by-side. In either configuration, the mounting scheme must allow for the center of gravity of the deployed boom to be on the geometric spin axis.

1.4.2 The spin plane booms (left side of figure 4)

- Deployed wire boom length greater than 40 m, stowed length less than 0.35 m (not including the sensor spheres).
- Capable of supporting two 10 cm diameter sensor spheres of 0.15 Kg each along the wire length, one at 75% radially out and the other at the tip.

- Wire bundle for power and signal equivalent to 0.03 kg/m linear density
- Boom and deploying mechanism less than 2 kg
- The attachment point at the spacecraft for the booms must be at 90° ±5° around the circumference of the spacecraft.
- The attachment points must be in a plane and that plane must contain the spacecraft center of gravity. The center of gravity may be up to 0.6 cm out of the plane.

1.4.3 Electronics Box

The electric field electronics box is estimated to weigh 2.5kg and be approximately 15 cm X 15 cm X 13 cm. It may be assumed that the Electric Field instrument has a Mil-Std-1553 or other high level interface.

1.5 Magnetometer Instrument

The magnetometer instrument consists of a sensor head and an electronics unit.

1.5.1 Spacecraft magnetic field

The spacecraft must not generate a DC magnetic field of more than 1nT at the sensor head and the spacecraft magnetic field must be known to 0.1nT DC & AC (up to 500Hz TBV). In light of these requirements it is up to the spacecraft manufacturer whether he uses a boom to mount the magnetometer head away from the main body of the spacecraft or if the magnetic cleanliness of the spacecraft is such that a boom is not necessary. The magnetometer head weighs 0.5 kg and is 31 cubic centimeters. In any event, the mass and cost of a boom will be borne by the spacecraft.

1.5.2 Magnetometer pointing accuracy

The spacecraft pointing requirements are based on the requirements of the magnetometer. The fundamental MMS pointing requirement is to be able to know the magnetometer sensor head position to 0.1°.

1.5.3 Co-alignment of Magnetometer and Electric Field booms

The alignment of the Magnetometer sensor head and the electric field booms must be known to 0.1° in the on orbit configuration.

1.5.4 Correlation of Magnetometer data and Electric Field data

The spacecraft timing system shall have sufficient resolution to allow correlation of electric field telemetry and magnetic field telemetry to less than 400μs. The time tag shall be applied by the C&DH system, upon receipt of data from the instrument.

1.5.5 Correlation of Magnetometers between spacecraft

The spacecraft C&DH subsystem shall allow correlation between spacecraft of magnetometer data. The IRAS shall allow relative time correlation of 400μs between spacecraft.

1.6 Interspacecraft Ranging and Alarm System

1.6.1 IRAS requirements

1. The IRAS shall measure the distance between the five Magnetospheric Multi-Scale spacecraft. The requirement is to measure the distance between the spacecraft with accuracy better than 1%.
2. A low speed serial message shall be passed from the IRAS to the spacecraft C&DH processor. The message should contain the following information: distance to other members of the formation, alarm status of each spacecraft, thruster firing status of each spacecraft, internal IRAS health and safety.

3. Maximum time from alarm message input to transmitting IRAS to alarm signal output from receiving IRAS shall be 3 sec.
4. The IRAS system shall be operational at all times in all mission modes, up to 2Re.
5. The IRAS system shall be capable of correlating time among the five spacecraft to less than 400µs

1.6.2 IRAS to C&DH interfaces

1.6.2.1 Telemetry and Alarms

The IRAS shall have a Mil-Std-1553 interface to allow low speed telemetry and high speed alarms to pass among the spacecraft.

Table 3

Alarm	C&DH response	Comments
Interesting science	Send message to all instruments within 0.5second of receipt	Other spacecraft can go into a high speed data capture mode on receipt of this alarm
One spacecraft has aborted a thruster firing	Receiving spacecraft abort thrusting	During maneuvers the five spacecraft must remain together. In the event that one of the spacecraft has to abort a maneuver, this message allows other spacecraft to abort also.

The Mil-Std-1553 schedule table shall allow for ‘interesting science’ messages to be passed from any of the instruments to the IRAS within 0.5s. The spacecraft C&DH shall pass a message to the IRAS in the event that that spacecraft has aborted a maneuver within 0.5 seconds.

1.6.2.2 Timing

The C&DH shall pass to the IRAS a discrete RS-422 timing pulse that is correlated with the fundamental timing system of the spacecraft. The pulse repetition frequency shall be 1Hz. The IRAS will use the pulses to correlate time among the spacecraft. The correlated time information shall be telemetered from the IRAS as Mil-Std-1553 messages. There is no requirement for the spacecraft to read or use the correlated time messages. The correlated time messages will be used on the ground to correlate time between the spacecraft instruments.

1.6.3 IRAS antennas

The IRAS will have two S-band antennas that are mounted to cover 4π steradian. The locations of the antennas will be negotiated between the spacecraft manufacturer and the IRAS designers.

Appendix E

MMS Mission Study

Period of Performance, Schedule and Price

Period of Performance, Schedule and Deliverables

The period of performance for this study shall be nominally 100 days after receipt of order (ARO).

Study Schedule, Meetings and Location

Date	Activity	Location	Duration
Within 7 days ARO	Kick-off Telecon	N/A	½ day
Within 50 days ARO	Mid-term Status Review	Contractor's location	1 day
Within 100 days ARO	Final Presentation	Contractor's location	1 day
As needed	Informal exchanges	Via telecon & e-mail	

Delivery Schedule

Date	Item	# of Copies
Within 90 days ARO	CAD model	1 on electronic media
Within 100 days ARO	Final report	5 hard copies and 1 on electronic media

Price

The NASA envisions two equal payments:

Payment 1 at completion of mid-term status review	$_tbp__
Payment 2 at NASA acceptance of Final Report	$_tbp__
Total	$_tbp__